BELL AEROSPACE COMPANY
DIVISION OF TEXTRON

MONTHLY PROGRESS REPORT NO. 6

FOR

HIGH PRESSURE REVERSE FLOW APS ENGINE

CONTRACT NO. NAS 3-14353

(NASA-CR-117950) HIGH PRESSURE REVERSE FLOW APS ENGINE Monthly Progress Report (Bell Aerospace Co.) 43 p

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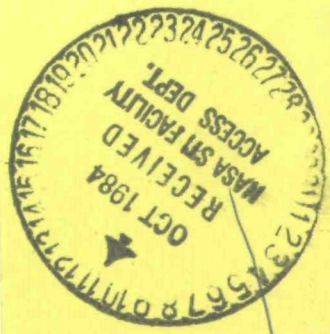

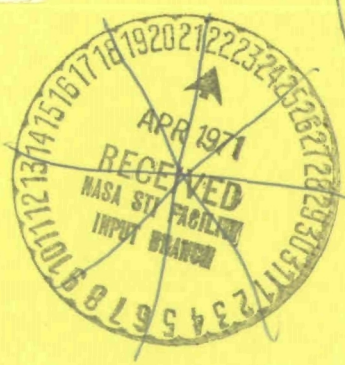

Bell Aerospace Company
A Division of Textron Corp.
Buffalo, New York 14240

BELL AEROSPACE COMPANY

Report No. 8636-933006

MONTHLY PROGRESS REPORT NO. 6

FOR PERIOD ENDING MARCH 31, 1971

HIGH PRESSURE REVERSE FLOW APS ENGINE
CONTRACT NO. NAS 3-14353

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I. INTRODUCTION

This report is submitted in compliance with the requirements of Contract NAS 3-14353 and summarizes the efforts performed on the High Pressure Reverse Flow APS engine demonstration program for the month of March 1971. The program includes analytical and experimental effort to investigate the use of gaseous hydrogen, gaseous oxygen propellants in an engine with a nominal thrust of 1500 lbs at a nozzle area ratio of 40:1.

The reverse flow thrust chamber consists of a spherical combustion chamber with a conventional nozzle. The oxidizer is injected into the combustion chamber from a single large diameter orifice at the conventional injection position. The fuel is injected upstream in the convergent portion of the nozzle and travels along the wall as a fuel film prior to being turned into the oxidizer at the oxidizer injection orifice. This upstream injection of the fuel "film" led to the term "Reverse Flow" to describe the propellant injection configuration. The reverse flow concept provides considerable simplification of propellant injection when compared to conventional multi-element injectors.

The principal objective of the program is the demonstration of high performance with a design capable of achieving long life. Specific goals include the achievement of 97% c* and a thrust chamber design demonstrating I_{sp} = 435 seconds and capable of 1 X 10^6 firings.

The reverse flow engine has several advantages relative to engines with conventional injector schemes:

1. The fuel and oxidizer are injected at different locations allowing a significant simplification in injector design. The injector joints are few in number, all are welded and all are fully inspectable.

2. The combustion chamber walls are film cooled with the total H_2 combustion flow. The externally insulated combustion chamber shell reaches steady state temperatures well within the capability of conventional materials of construction.

3. The relatively low temperature single wall combustion chamber allows flexibility for the installation and positioning of the ignition system.

4. The nozzle section liner longitudinal thermal stresses are independent of the combustion chamber improving the life cycle capability of the liner.

The program is scheduled for completion in August 1971 and the effort is divided into the following tasks:

Task I - Injector Analysis and Design

Task II - Injector Fabrication

Task III - Thrust Chamber Analysis and Design

Task IV - Thrust Chamber Fabrication

Task V - Ignition System Analysis and Design

Task VI - Ignition System Fabrication and Checkout

Task VII - Propellant Valve

Task VIII - Injector Tests

Task IX - Thrust Chamber Cooling Tests

Task X - Pulsing Tests

Task XI - Reporting Requirements

Summary - Results to Date

1. The effort in March concluded the testing in the sea level test cell. Drawings of thrust chamber designs for altitude testing were completed and hardware fabrication was initiated.

2. High performance (% c*) has been demonstrated at heat rejection rates consistent with operation for one million firing cycles of essentially unlimited duration.

II. TASK REPORT

A. Task I - Injector Analysis and Design

No effort was performed under this contract task during the month of March 1971. Related company sponsored effort of reverse flow engine injector design and analysis is described in the Supplement to this report.

B. Task II - Injector Fabrication

There was no additional expenditures for injector fabrication during the report period.

C. Task III - Thrust Chamber Analysis and Design

The design of detail and assembly drawings of the basic hardware for the thrust chamber cooling tests at simulated

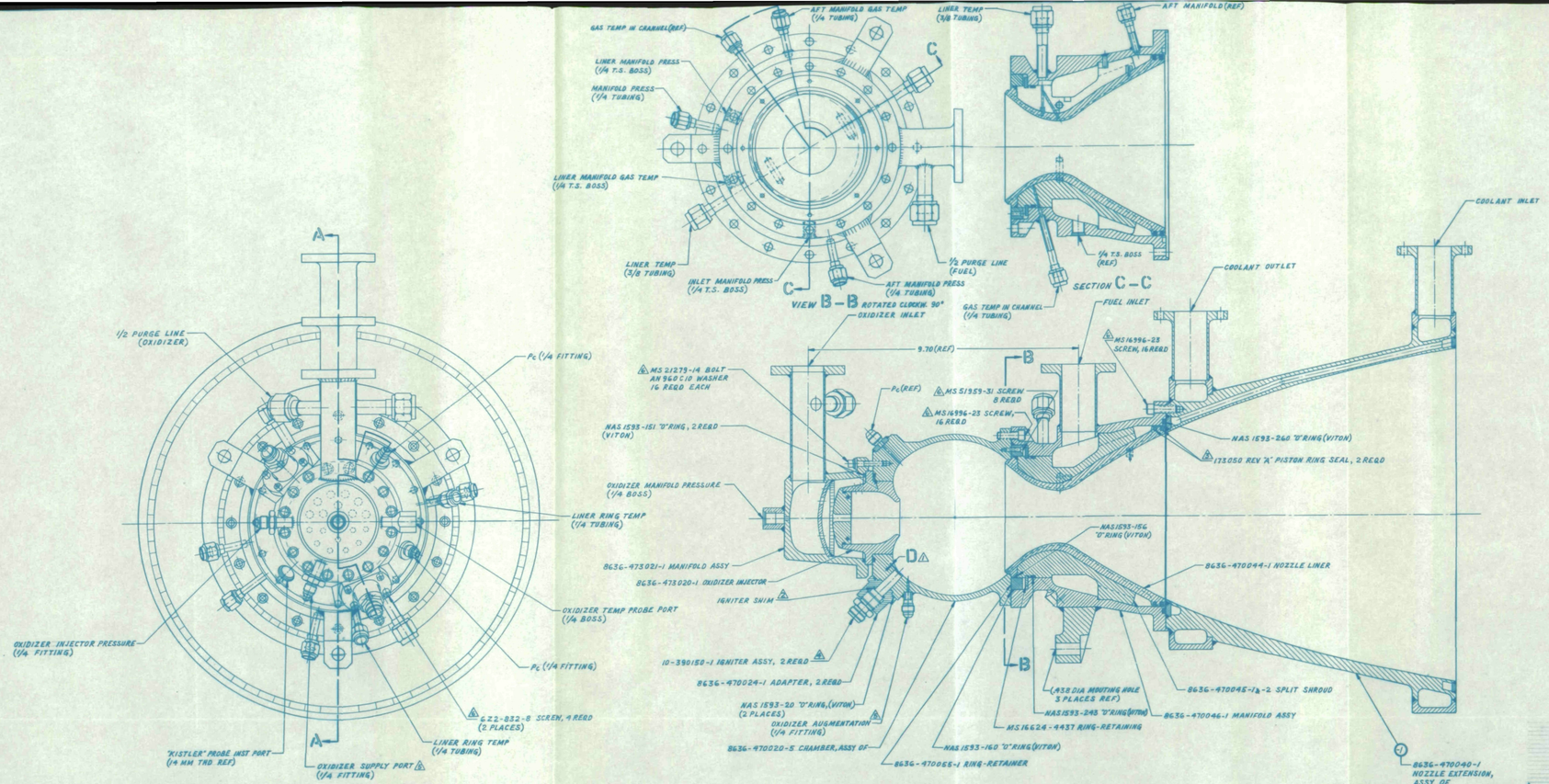
altitude conditions were completed. The effort included the regenerative cooled nozzle subassembly, the water/H_2 cooled nozzle extension and the columbium nozzle extension. The hardware was described in general terms in the last monthly progress report. The remaining items of hardware, a nozzle section liner providing dump cooling of the nozzle extension and a liner for throat section film cooling will be released in the next report period.

The assembly drawing of the engine with the cooled nozzle extension is presented in Figure 1.

The nozzle extensions include an optimized 75% bell configuration. The columbium extension will be fabricated from 103 alloy with a Sylcor R512E (Sylcor Division of Sylvania Electric) oxidation resistant coating.

The piston rings for the expansion joint of the nozzle section were ordered from Precision Piston Ring, Inc. The rings are a 3 piece configuration fabricated from Haynes 25 steel. The mating surface of the nozzle section stainless steel jacket will incorporate a No. 16 finish to provide a seal with the rings.

Thermocouples were included in the nozzle and extension drawings. The installation includes cromel alumel couples with short sections of 1/16 inch. diameter sheaths. The thermocouples will be peaned into the liner lands and the cooled nozzle extension to provide temperature data to establish the gas side wall temperatures.



ITEM	QUAN	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	MATERIAL OR SPECIFICATION	MATERIAL OR NOTE	DRAWING NO	LIST
2		173050 REV "X"	PISTON RING SEAL				
2		10-390150-1	IGNITER ASSY				
1		NAS1593-260	"O" RING			STD	
1		NAS1593-248	"O" RING			STD	
1		NAS1593-160	"O" RING			STD	
1		NAS1593-156	"O" RING			STD	
2		NAS1593-151	"O" RING			STD	
2		NAS1593-20	"O" RING			STD	
8		6Z2-832-8	SCREW			BELL STD	
16		AN960C10	WASHER			STD	
1		MS16624-4437	RING-RETAINING			STD	
8		MS51959-31	SCREW			STD	
16		MS16996-23	SCREW			STD	
16		MS21279-14	BOLT			STD	
1		8636-473021-1	MANIFOLD ASSY				
1		8636-473020-1	OXIDIZER INJECTOR				
1		8636-470055	RING-RETAINER				
1		8636-470046-1	MANIFOLD ASSY				
1		8636-470045-SPLIT	SHROUD				
1		8636-470045	SPLIT SHROUD				
1		8636-470044	NOZZLE LINER				
1		8636-470040	NOZZLE EXTENSION, ASSY OF				
2		8636-470024-1	ADAPTER				
1		8636-470020-5	CHAMBER ASSY OF				
1		8636-470050	ALTITUDE ENGINE ASSY				

PARTS LIST

NOTES:

1. TORQUE AS REQD PER ST1001-947008.
2. OXIDIZER SUPPLY PORT SHALL BE CONNECTED TO OXIDIZER AUGMENTATION PORTS WITH A 1/8 TUBE ASSY. LENGTH & CONFIGURATION TO BE DETERMINED BY MOCK UP.
3. PURCHASE FROM "BENDIX CORP., SCINTILLA DIV, SIDNEY, N.Y.
4. PURCHASE FROM "PRECISION PISTON RINGS, INC. INDIANAPOLIS, INDIANA.
5. THICKNESS OF IGNITER SHIM TO BE DETERMINED BY DIMENSION "D".
6. DIMENSION "D" TO BE DETERMINED BY ENGRG.

SIZE	CODE IDENT NO	ALTITUDE ENGINE ASSY	
	80070	8636-470050	

FIGURE 1

8636-470050

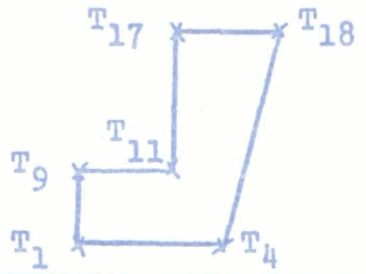
The detailing of the zirconium copper nozzle section liner cooling passages included additional two dimensional thermal analyses to size the coolant passages for a maximum combustion side wall temperature of 750°F nominal at a chamber pressure of 300 psia, O/F of 4.0 and a propellant inlet temperature of 540°R. After initial definition of passage height and width as a function of axial distance along the nozzle section, a ten time size layout drawing was constructed. Passage geometries were modified to conform to feasible manufacturing processes and conventional design practices. The new passage configuration was then reanalyzed at nominal temperature (540°R) and at the extremes of operation, 200°R and 800°R. The results of these analyses are presented in Table I.

The data of Table I shows that the nominal temperature of 750°F was approached closely at 540°R H_2 inlet temperature. The maximum temperatures for low and high propellant feed temperatures are 727°R and 970°R, respectively.

The projected coolant temperature rise of 128°F under nominal operating conditions obtained from the heat transfer analysis was used to size the injector for the selected injection velocity of 2400 feet/sec. The coolant temperature rise at mixture ratios of 3 and 5 were calculated by extrapolation based upon test data obtained from a 1K regeneratively cooled engine of a similar configuration. Table II presents the expected operating conditions of the regeneratively cooled nozzle with and

TABLE I

SUMMARY OF NODAL TEMPERATURES
WITH VARIOUS COOLANT INLET TEMPERATURES



	0.4 Aft of Throat Max. Temperature Region			0.4 Fwd of Throat Region Max. Heat Flux			1.05 Fwd of Throat		
T_L, °R	540	200	800	540	200	800	540	200	800
T_1	790	727	970	674	487	882	741	591	940
T_4	779	715	959	941	452	851	718	565	919
T_9	686	623	870	530	339	744	603	451	808
T_{11}	652	586	839	492	295	710	561	400	770
T_{17}	466	385	667	277	46	513	397	216	620
T_{18}	482	403	682	293	65	528	417	238	638

T_L = Inlet Coolant Temperature, °R

T_1, T_4, T_9, T_{11}, T_{17}, T_{18}, Nozzle Temperature, °F

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TABLE II

REGENERATIVELY COOLED NOZZLE DESIGN DATA

	NO DUMP COOLING			6% DUMP COOLING		
MIXTURE RATIO	3	4	5	3	4	5
Hot - Steady State						
Injection Velocity ft/sec	2734	2400	2155	2724	2396	2156
Propellant ΔT ^{o}F (Total Temp)	91	128	150	97	136	160
Cold (Transient)						
Injection Velocity ft/sec	2362	1958	1701	2333	1933	1678

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without 6% dump cooling at an area ratio of ten. A constant injection velocity of 2400 ft/sec. is maintained for both concepts by changing (reducing) the fuel injection crossectional area for the condition with dump cooling.

Analysis is currently being conducted to evaluate 6% film cooling in the convergent section of the nozzle, and to determine the effect of coolant passage surface finish on injector pressure drop. A preliminary analysis of the pressure drop for the all-regeneratively cooled nozzle is shown in Figure 2.

In order to insure a free-standing nozzle under all operating conditions, a two dimensional thermal expansion analysis was also conducted based on the detailed liner configuration. The temperature used for this analysis was taken directly from the two-dimensional heat transfer analysis previously described. It was found that with the nozzle fixed at the injector end that the thermal growth (combined radial and axial) in the divergent section was along the contour, and indicated a maximum in gap of .005" between the nozzle and the shroud (see Figure 3). The convergent section requires a maximum gap of .013" at a distance of 1.75 inches upstream of the throat, and .008" at the geometric throat centerline. An additional analysis was conducted to determine the radial growth of the nozzle flange at the injector end where it is held fixed. Due to radial growth of the flange at this location a radial clearance of .007 inches was incorporated to prevent permanent yielding during operation.

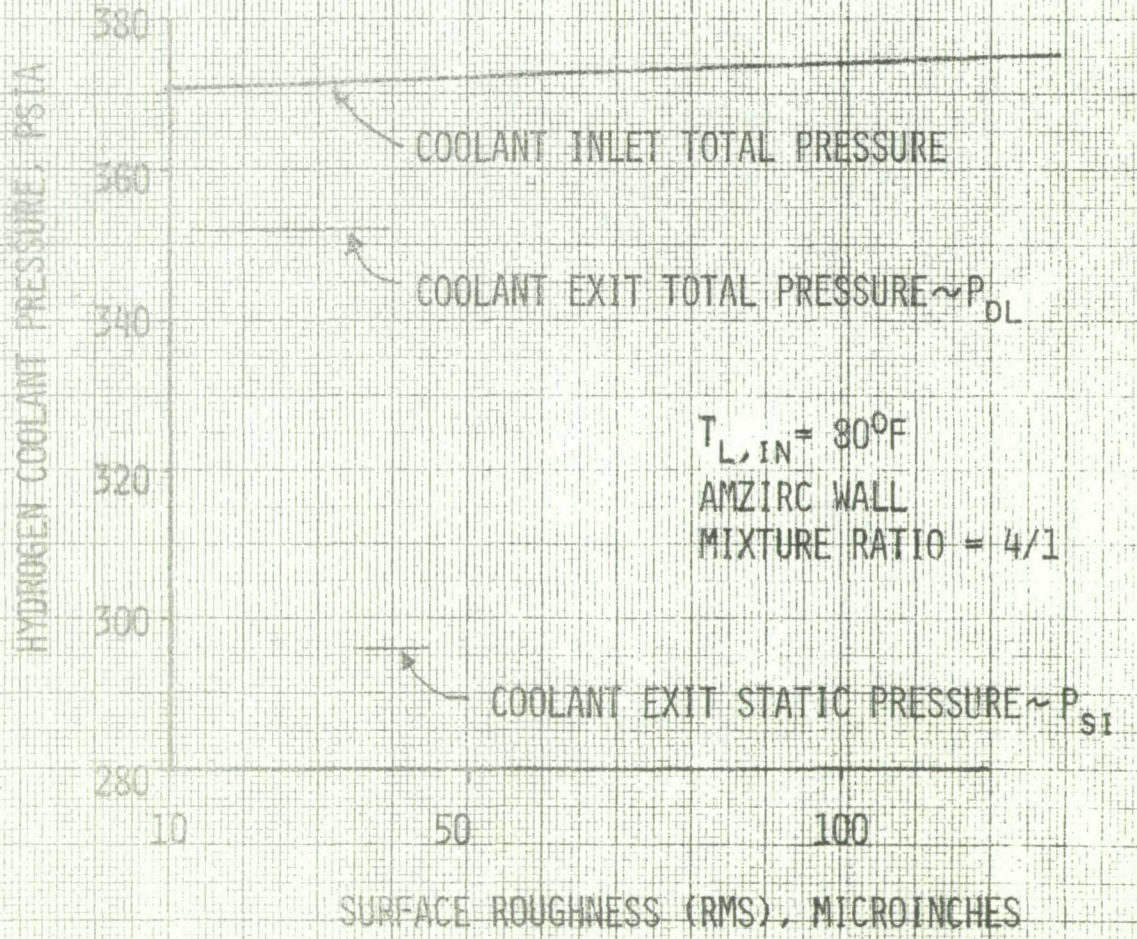
FIGURE 2

COOLANT PASSAGE PRESSURE LOSS

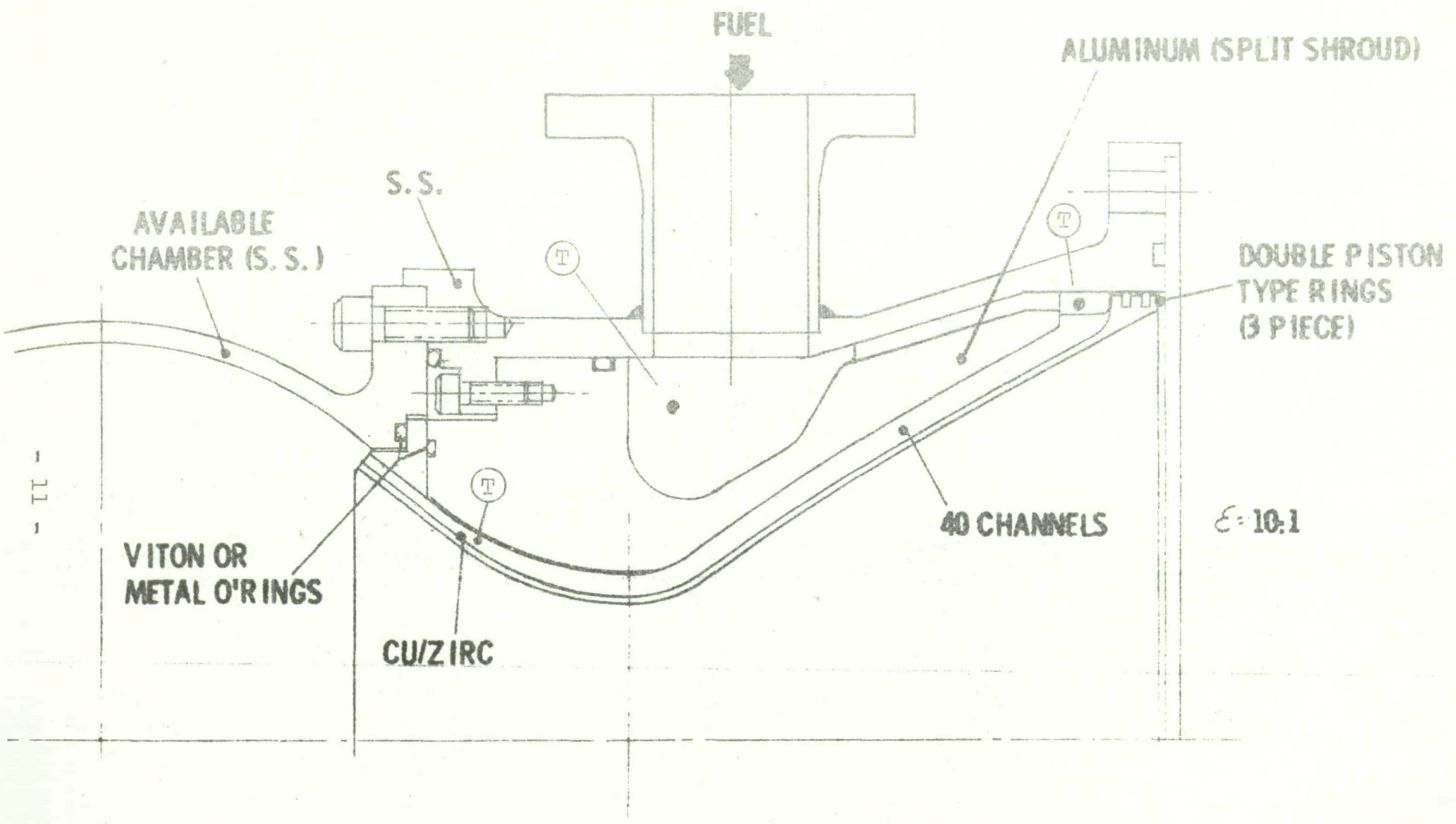
VERSUS

CHANNEL SURFACE FINISH

ALTITUDE NOZZLE



K+E 10 X 10 TO THE CENTIMETER 46 1513
KEUFFEL & ESSER CO. MADE IN U.S.A.



1500 LBF COOLED NOZZLE SECTION (REGENERATIVE COOLED)

FIGURE 3

D. Task IV - Thrust Chamber Fabrication

Fabrication of the regenerative cooled nozzle sub-assembly and the cooled nozzle extension were initiated. Material was purchased for the columbium nozzle extension.

E. Task V - Ignition System Analysis and Design

No changes were made to the previously established ignition system.

F. Task VI - Ignition System Fabrication and Checkout

Fabrication of the ignition system to support altitude level tests continued. Delivery of completed hardware is scheduled during next report period. Ignition characteristics are being evaluated under company support as discussed in the appendix of this report.

G. Task VIII - Propellant Valve

The gas operated oxidizer and fuel ball valves continued in operation for injector testing. The valves have shown no change in operating characteristics which includes 14 ms opening and closing times, zero leakage and approximately 10 psi ΔP at rated flows.

H. Task VIII - Injector Tests

The injector testing with the heat sink hardware was completed with high P_c and low P_c firings. The tests conducted during the report period are summarized in Tables III and IV. A total of 37 firing tests were made. Instrumentation recording

TABLE III

INJECTOR CONFIGURATIONS TESTED

L*	OX. α	FUEL $\mathcal{M}$	% CENTERFLOW	SWIRL CUP D_2	NOMINAL P_c
32	4.35	.50	5	1.189	300
32	5.22	.50	5	1.427	300
32	5.22	.50	10	1.427	300
32	6.44	.50	10	1.427	300
32	6.44	.50	15	1.427	300
32	6.44	.56	10	1.427	300
32	6.44	.56	10	1.427	500
32	6.44	.56	15	1.427	300
32	6.51	.56	5	1.189	300
32	6.51	.56	5	1.189	100
32	9.22	.56	10	1.427	300

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TABLE IV

	HARDWARE							TEST PERFORMANCE					DATA CORRECTED TO 540°R					
Run No.	Date	Cup	Inj.	Noz.	Cap.	Cham.	W_F	W_O	MR	Pc	C*	F	Isp	C*	Isp SL	CF SL	Isp Vac	CF Vac	%C*	%Isp Vac
1092	3-5-71	-3	029-1	Inte. with Inj.	-3	-020	.705	2.701	3.83	298.8	7772	1096	321.9	7808	325.0	1.3392	353.4	1.4560	94.5	91.5
1094																				
1095							0.564	2.924	5.18	296.9	7539	1093	313.5	7570	316.2	1.3441	343.9	1.4617	95.7	92.6
1096	3-8-71	-5					0.772	2.800	3.88	306.1	7575	1118	317.5	7716	320.8	1.3377	348.1	1.4517	93.5	90.3
1098							0.547	3.031	5.54	301.7	7452	1114	311.3	7480	313.9	1.3503	340.8	1.4659	95.9	92.9
1100					-1		0.667	2.858	4.29	304.8	7629	1112	315.5	7664	318.5	1.3372	345.8	1.4517	94.0	90.5
1102							0.673	2.683	3.99	293.3	7711	1066	317.6	7744	320.6	1.3319	349.2	1.4509	94.2	90.8
1103							0.542	3.141	5.80	303.3	7266	1113	302.2	7293	304.8	1.3444	330.8	1.4595	94.4	91.0
1106	3-12-71	-1B					0.691	2.754	3.99	301.3	7732	1095	317.9	7768	321.0	1.3294	348.9	1.4450	94.4	90.7
1107							0.584	2.928	5.01	301.1	7565	1096	312.1	7595	314.8	1.3337	342.2	1.4496	95.5	91.6
1109	3-12-71						0.706	2.740	3.88	299.5	7669	1084	314.5	7705	317.5	1.3260	345.4	1.4425	93.4	89.6
1110	3-12-71						0.583	2.978	5.11	301.3	7467	1094	307.3	7497	310.0	1.3304	337.0	1.4462	94.6	90.5
1111	3-15-71	-1B	014-9	017-1	019-9		0.856	3.054	3.57	303.7	7858	1266	323.8	7889	326.8	1.3327	364.0	1.4846	94.9	90.5
1113							0.628	3.300	5.26	293.6	7562	1222	311.1	7588	313.8	1.3307	350.9	1.4878	96.3	90.9
1114	3-16-71						0.772	3.034	3.93	298.1	7891	1228	322.6	7922	325.6	1.3224	363.7	1.4772	96.2	91.0
1115							0.627	3.333	5.32	297.5	7568	1231	310.9	7597	313.7	1.3287	350.4	1.4838	96.6	90.9
1121	3-22-71	-3A			019-3		0.733	2.635	3.60	269.7	8037	1094	325.0	8076	328.4	1.3082	371.3	1.4793	97.2	92.3
1122							0.816	2.893	3.54	297.2	8040	1221	329.1	8079	332.5	1.3240	372.1	1.4816	97.0	92.0
1123							0.722	3.161	4.38	305.3	7888	1261	324.6	7922	327.8	1.3313	365.0	1.4824	97.4	92.2
1125																				
1[illegible]	3-23-71						0.640	3.374	5.27	307.2	7714	1279	318.7	7743	321.6	1.3363	357.7	1.4865	98.3	92.7
1127	3-23-71	-1B					0.665	2.794	4.20	307.8	7856	1121	324.1	7888	327.2	1.3345	363.9	1.4843	96.5	91.5
1128							0.612	2.850	4.66	303.3	7735	1104	318.9	7767	321.9	1.3336	358.6	1.4857	96.4	91.3
1129	3-24-71	-9					0.808	2.816	3.49	283.6	7920	1167	321.9	7955	325.0	1.3146	365.2	1.4773	95.5	90.7
1130							0.672	3.208	4.77	292.3	7624	1204	303.0	7655	313.2	1.3166	350.8	1.4744	95.4	89.5
1132	3-25-71	-3A			019-3		0.154	.732	4.76	68.2	7902	187	211.1	7933	214.0				98.8	
1133							0.329	1.209	3.68	122.1	8152	421	273.8	8196	277.2				98.9	
1134							0.282	.975	3.46	100.3	8197	321	255.5	8243	258.9				99.0	
1135							0.226	1.188	5.24	106.9	7765	356	252.1	7797	255.0				98.8	
1136							0.804	2.918	3.63	295.0	8000	1215	326.3	8038	329.6	1.3193	368.7	1.476	96.8	91.7
1137		-1B			019-1		0.668	2.777	4.16	290.9	7725	1105	320.8	7755	323.8	1.3433	362.0	1.502	94.8	91.0
1138							0.577	2.965	5.13	292.3	7551	1111	313.8	7580	316.7	1.3442	353.9	1.502	95.7	91.3
1139							1.170	3.852	3.29	443.5	8080	1735	345.5	8128	349.3	1.3826	375.5	1.487	97.3	93.0
1140							0.947	5.716	6.03	517.2	7470	2155	323.4	7497	326.2	1.4234	351.2	1.533	97.9	93.5

system failures resulted in a loss of data for 3 firings. The loss of data resulted from procedural and equipment failures which were corrected.

Mass flowrate data reduction was reviewed on the basis of flow calibration and reduction procedures recommended by R.E. Johnson of Lewis Labs. All tests performed to date for the 1500 lb injector testing used swirlmeter to measure gaseous propellant flow rates. These swirlmeters were originally calibrated against a 0.796 inch sonic nozzle. Calibration of this master nozzle was performed by the Colorado Engineering Experiment Station, Inc. (CEESI). These CEESI discharge coefficients were used in the flow equation to determine swirlmeter calibration factors.

There was no reason to question the CEESI calibration until it was learned in a visit to a NASA test facility that a possible discrepancy existed between the computed discharge coefficient and that obtained from the raw data. A spot check of CEESI computed discharge coefficients using a theory which NASA considered appropriate, did not check with the raw data. As a result, discharge coefficients were computed for the sonic nozzle and deviations were found (up to 1.5% at high Reynolds number) from CEESI computed values. The swirlmeter calibration factors were recomputed using NASA/Johnson computed nozzle discharge coefficients and reduction procedures.

A few of the heat sink hardware firings had been re-reduced at the end of the report period on a basis of R.E. Johnson's work and data changes were indicated. Minor shifts of mixture ratio were calculated; the derived c* values generally increased from 0 to 1% relative to data reported earlier. Specifically, the c* data reported for the α 10.73, fuel injection Mach number of 0.56 and 5% center flow remained essentially unchanged, at an O/F of 4.0 but the c* of the α 6.51, M 0.56, 5% center flow configuration increased approximately 1%. The α 6.44, M 0.56, 10% center flow showed a similar increase. The data comparison is presented in Figure 4. The falloff of % c* of the revised data for the α 10.73 configuration is more consistent with the operation of that cup with lower design Mach number fuel injection as described in the last monthly progress report. Additional run data will be recalculated during the next report period. With the exception of Figure 4, the remaining data in this report (including Table IV) does not include the corrected flow data.

The test firings during March completed the evaluation of several configurations tested earlier (the α 4.35, 5.22, and 6.44) by providing additional data points particularly at the high O/F range, and further assessment of the operation with higher percentages of center flow. The tests with increased centerflow were made to improve the characterization of that parameter on the combustion efficiency in view of the reduction of oxidizer pressure drop with increased % centerflow.

The attempts to demonstrate higher % c* was continued with relatively high oxidizer ΔP configurations, the α 6.51 and 9.22 assemblies. The α 6.51 configuration utilized one extreme

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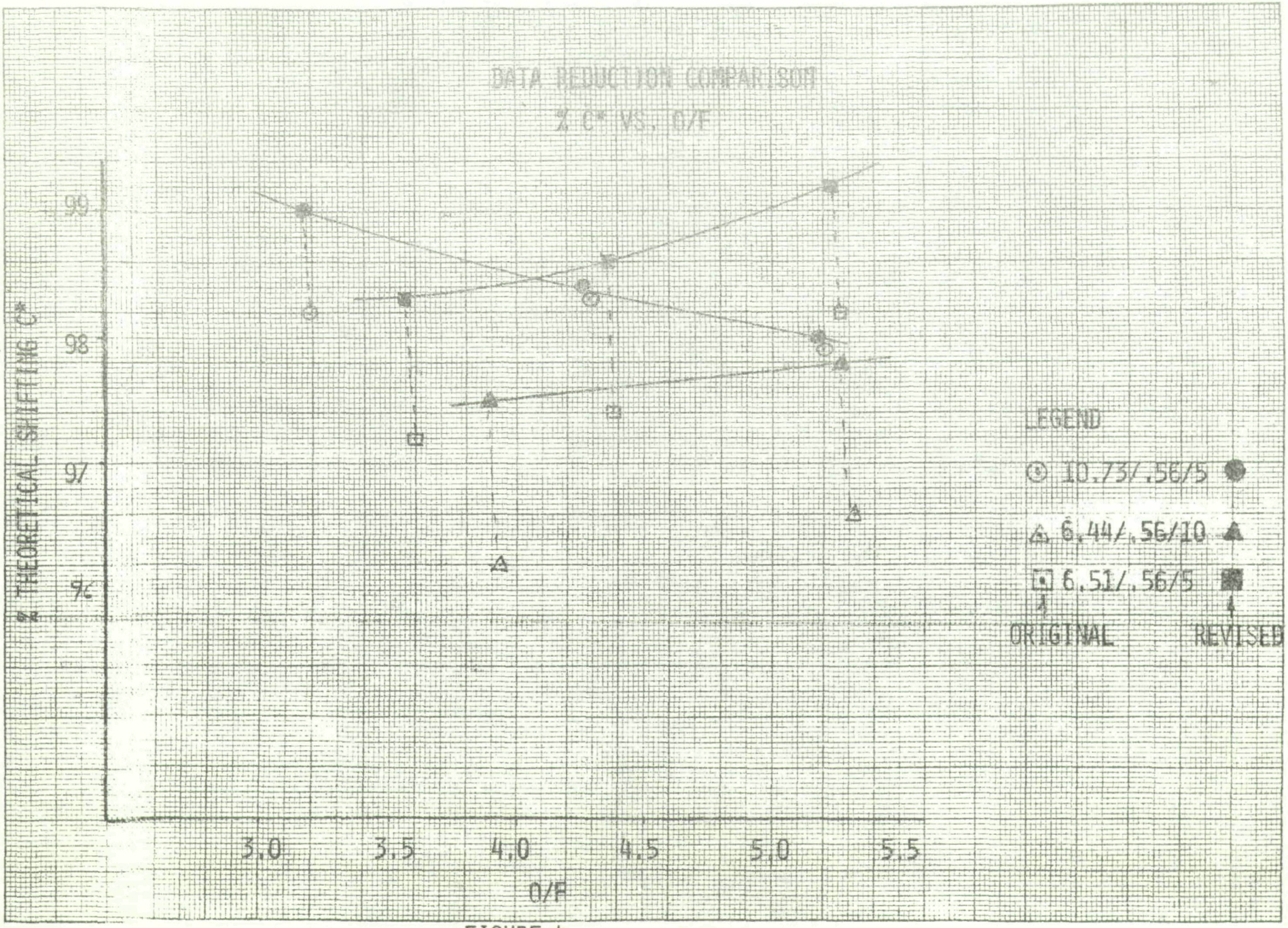


FIGURE 4

of the cup outlet diameter, D_2 = 1.189 inches, and the 9.22 α represented a variation to the 10.73 cup discussed in the last monthly progress report. The % c* of the 6.51 exceeded the program goal of 97% at a mixture ratio of 4.0. The lower performance of the α 9.22 configuration does not follow the general pattern of increasing %c* with increasing α.

The α 6.51 configuration was selected for low P_c operation. The % c* increased at a low P_c as shown in Figure 5. The α 6.44 oxidizer cup was employed for the high P_c firings because of the high pressure drop of the 6.51 for operation at 500 psia. The high P_c data are included in Figure 5. Same performance improvement was indicated at increased P_c.

The selection of the oxidizer injector to be employed for altitude testing includes a consideration of heat rejection pressure drop and % c*. The higher performance of the α 6.51 and 10.73 configurations is achieved at the expense of increased ΔP and heat rejection as shown in Table V. The tentative selection of the α 6.44 cup with 10% center flow is based on its lower ΔP and heat rejection. The re-reduction of the 6.44 configuration indicates a % c* which exceeds the contract goal of 97%.

The significance of the Q/A data is further indicated by Figure 6, which presents a plot of cumulative structural damage, N, as a function of maximum gas side temperature of the .070" thick zirconium copper flloy selected for the regeneratively cooled thrust chamber assembly hardware. The cumulative damage analysis methods were reported in some detail in the last

LOW AND HIGH P_c PERFORMANCE

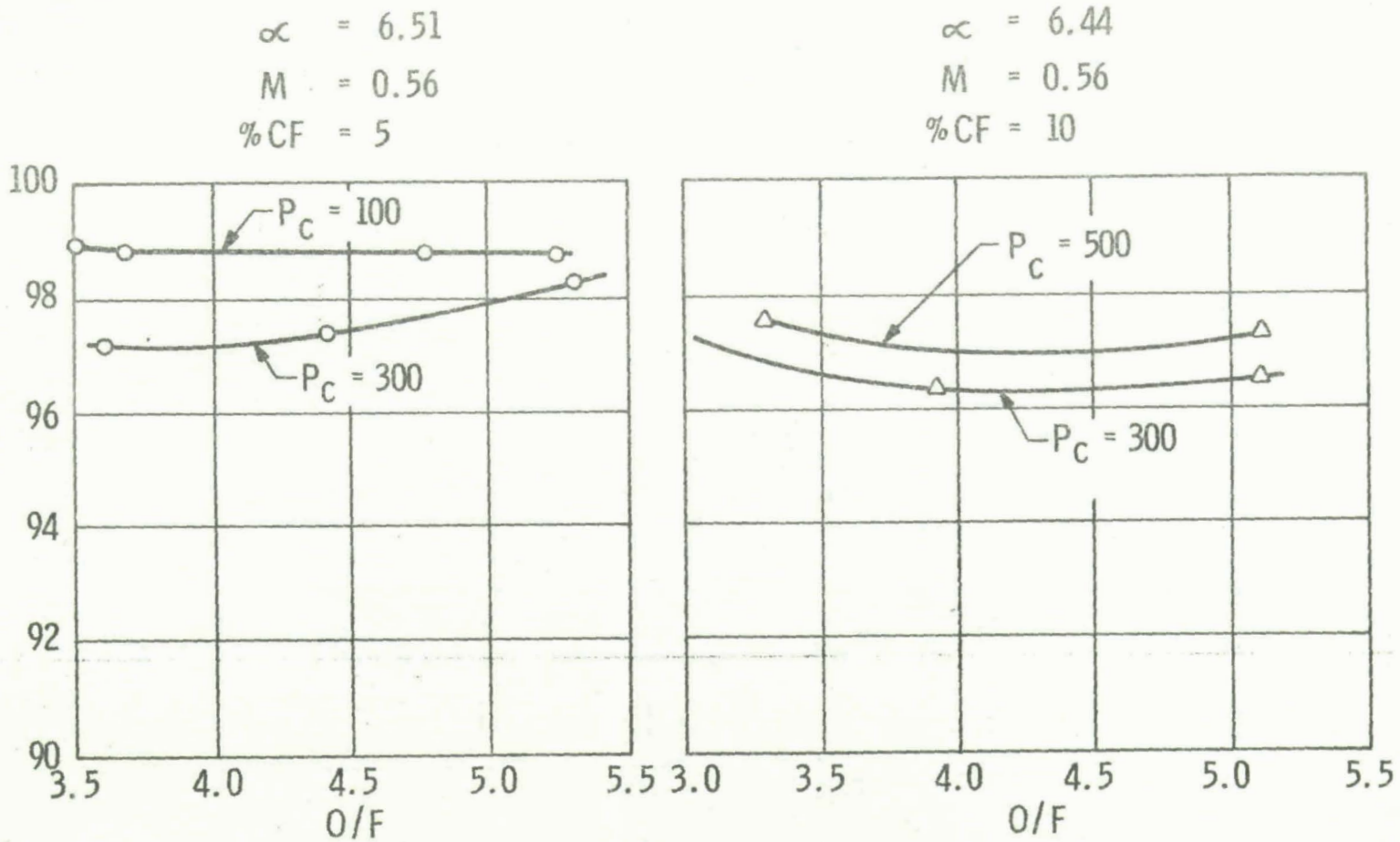


FIGURE 5

TABLE V

FUEL AND OXIDIZER INJECTOR COMPARISON

OXIDIZER α	OXIDIZER % CENTERFLOW	FUEL MACH NUMBER	η_{c^*} @ O/F = 4.0 P_c = 300	ΔP_{ox} HEAT SINK HARDWARE	PROJECTED ΔP_{ox} FLT. CONF.	Q/A THROAT HEAT SINK HARDWARE
10.73	5	0.56	>98%(1)	218	150	13.7
10.73	10	0.56	>97%(1)	180	130	12.0
6.51	5	0.56	>98%(1)	247	170	13.3
6.44	5	0.56	>96%	133	100	12.0
6.44	10	0.56	>97%(1)	85	75	11.0

ΔP Fuel at MN = 0.56 projects to less than 90 psi with regenerative cooled nozzle.

(1) Re-reduced data per Johnson calibration technique.

BELL AEROSPACE COMPANY

FREE STANDING AMZIRC NOZZLE LINER CUMULATIVE DAMAGE VERSUS T WALL GAS

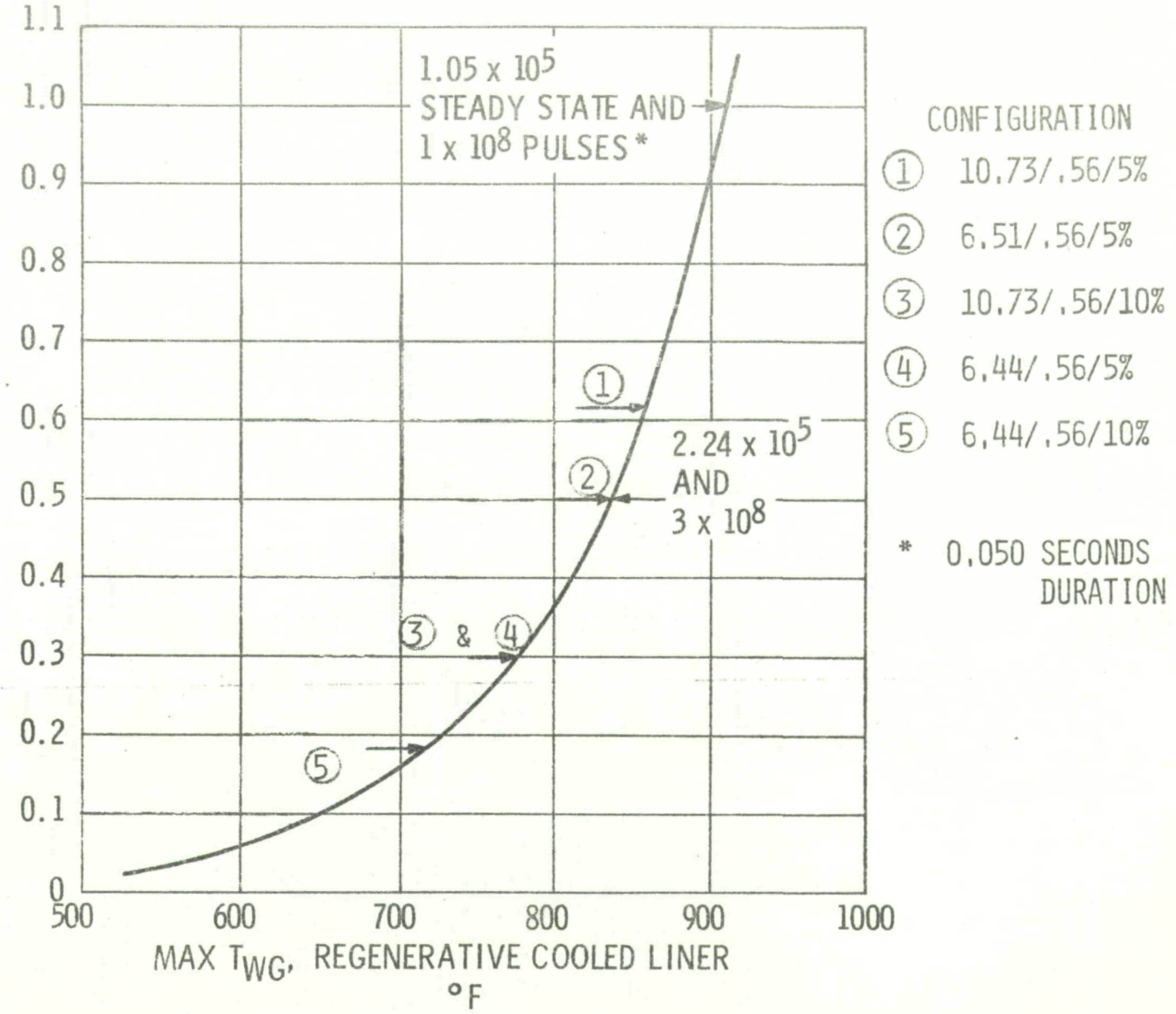


FIGURE 6

quarterly progress report. The references to the calculated values of pulse and steady-state firings for N = 1 and N = 0.5 shows that the design is limited by the number of steady-state rather than pulse firings. However, the calculations of two dimensional temperature-time transients for the throat liner and the resultant thermal and pressure stresses indicate that pulse operation, (operation to a stress level substantially less than that for long duration firings) is associated with very short duration pulses; 50 milliseconds or less.

The pulse width selected for the calculations was 50 milliseconds. Because that pulse duration may be unrealistically short, and because essentially steady-state stresses are incurred for firings of .10 to .20 seconds duration and greater, the engine operation capability should be directed toward values of N of Figure 6 of not greater than 0.1 to provide 1 X 10^6 steady-state firing capability.

In other words, the contract requirement of 100,000 steady state firings and 900,000 pulses translates to one million "steady state" firings because the actual hardware achieves "steady state" in 0.10 to 0.20 seconds. N was defined as;

$$N = \frac{100,000}{NFS} + \frac{900,000}{NFP} + \frac{50}{t\sim}$$

where NFS is the calculated number of steady state cycles the thrust chamber can withstand,

NFP is the calculated number of pulse cycles the thrust chamber can withstand

$t\sim$ is the time to creep failure

At low values of T_{wg} NFP is much greater than the required 900,000 pulse cycles and $t\sim$ is >1000 hrs. Therefore;

$$N \approx \frac{100,000}{NFS} \quad \text{or } N = .1 \text{ for } NFS = 1 \times 10^6$$

The figure also shows the wall gas calculated from the firing tests of some of the oxidizer/fuel injector configurations listed. The T wall gas values included the definition of a Q/A from the transient thermocouple data of the heat sink hardware firings and calculation of the regenerative cooled liner temperature corresponding to those Q/A values.

The low $T_{\text{wall gas}}$ predicted for the 6.44/.56/10% centerflow configuration insures that a cumulative damage factor of less than 0.1 can in fact be achieved with the reverse flow engine. Some film cooling of the nozzle liner is attractive to reduce the fuel side ΔP of the engine. That film cooling should also reduce the $T_{\text{wall gas}}$ below that for $N = 0.10$ including any inaccuracies associated with the calculated values of $T_{\text{wall gas}}$ from the firing tests and the calculation of N as a function of T_{wg} for the selected liner design. The capability to operate for a million firings independent of firing duration would provide a most significant achievement and that capability is undoubtedly attainable with the reverse flow engine concept.

I. Task IX - Thrust Chamber Cooling Tests

The Task IX test plan was completed and forwarded to NASA Lewis early in the report period.

J. Task X - Pulsing Tests

No effort during the report period.

K. Task XI - Reporting Requirements

1. The Task IX Thrust Chamber Cooling Test Plans was submitted.

2. The second quarterly progress report was forwarded.

3. A presentation of the program progress was prepared for the "Space Transportation System Propulsion Technology Conference", scheduled for April 6-7, 1971. A draft copy was forwarded to the NASA Project Manager.

III. PLANNED EFFORT

Thrust chamber cooling test hardware fabrication will continue through the next report period. Engine buildup for the first scheduled altitude tests with the regenerative cooled nozzle section (without film or dump cooling) and the cooled nozzle extension will be initiated. The drawings for the film and dump cooling nozzle section liners will be completed.

The space shuttle engine altitude test facility (test cell 1BN Bell Test Center) will be brought to operational status. The cell checkout will include the heat exchangers installed for low and high temperature O_2 and H_2 gas conditioning.

IV. GOVERNMENT/FURNISHED PROPERTY USAGE

Forecast for consumption of Government Furnished Propellants - NAS 3-14353 is shown in Table VI.

TABLE VI. GFP PROPELLANTS FORECAST

	LO_2 (lb x 10^3)		Isopropanol (gas x 10^3)		Helium (sfc)	
	Forecast	Actual	Forecast	Actual	Forecast	Actual
August 1970	0	0	0	0	0	0
September 1970	0	0	0	0	0	0
October 1970	0	0	0	0	0	0
November 1970	0	0	0	0	0	0
December 1970	0	0	0	0	300	375
January 1971	0	0	0	0	300	325
February 1971	0	0	0	0	0	250
March 1971	0		0		0	250
April 1971	112		9.75		0	
May 1971	215		18.75		300	
June 1971	215		18.75		300	
July 1971	100		9.00		300	

V. CONTRACT PROGRESS SCHEDULE

The contract progress schedule has been updated to reflect the effort performed during the month. This updating includes the increase in time spent in Task VIII, Injector Tests. Other program items such as the cooled chamber (Tasks III and IV) and the testing of this chamber (Tasks IX and X) were not effected by the Task VIII extension. A two week change in the start of altitude testing is currently anticipated.

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Lewis Research Center

CONTRACT PROGRESS SCHEDULE

REPORT FOR MONTH ENDING: 2 April 1971

FORM APPROVED. BUDGET BUREAU NO. 104-R0007

9. NASA Use Only

a. NASA CODE

b. PROJECT MGR.

c. EVALUATION DATE

d. EXCEPTION CATEGORY

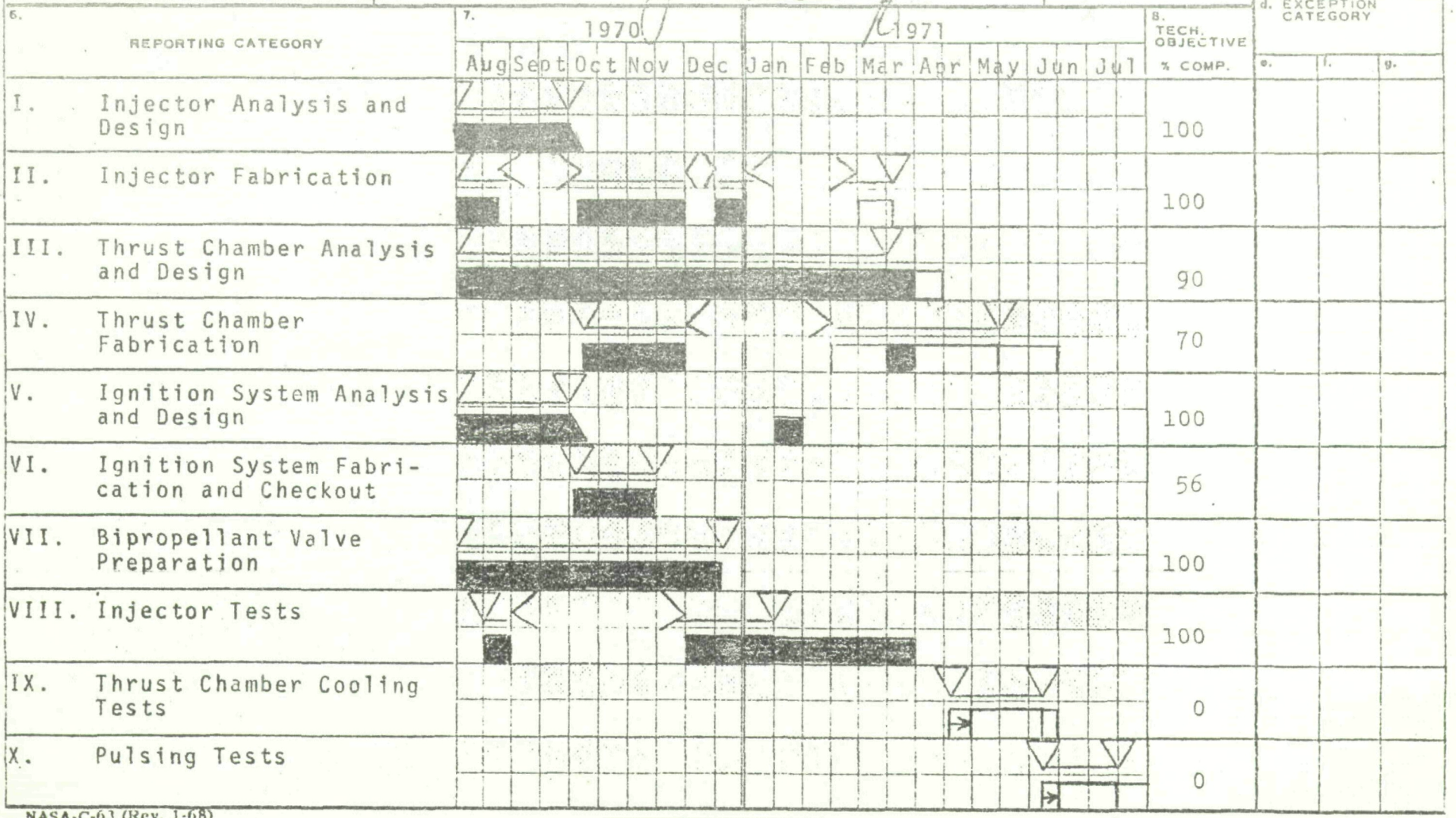
1. CONTRACT TITLE: High Pressure Reverse Flow APS Engine

2. CONTRACTOR (Name and address): Bell Aerospace Company, P.O. Box 1, Buffalo, New York 14240

3. CONTRACT NO.: NAS3-14353

4. APPROVED (Contractor's Project Manager): John M. Senneff — PREPARATION DATE

5. NASA APPROVED SCHEDULE DATE



NASA-C-63 (Rev. 1-68)

NASA APPROVED SCHEDULE

CONTRACTOR'S WORKING SCHEDULE

Symbology Table

S — NASA Selection

A — NASA Approval

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Lewis Research Center

CONTRACT PROGRESS SCHEDULE

REPORT FOR MONTH ENDING: 2 April 1971

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d. EXCEPTION CATEGORY

6. REPORTING CATEGORY	7. Aug	Sept	Oct	Nov	Dec	Jan	Feb	Mar	Apr	May	Jun	Jul	8. TECH. OBJECTIVE % COMP.	e.	f.	g.
XI. Program Plan & Test Plans													83			
XII. Monthly Reports													63			
XIII. Quarterly Reports													67			
XIV. Final Report Draft	7.								Jul	1971 Aug	Sept	Oct	0			

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NASA APPROVED SCHEDULE
CONTRACTOR'S WORKING SCHEDULE

Plans
- Program
- 1 Cold Flow Test
- 2 Injector Test Preliminary
- 3 Injector Test Final
- 4 Thrust Chamber Cooling Test

APPENDIX

Related company sponsored gaseous O_2/H_2 Reverse Flow Engine Development in progress in March 1971 or planned in the near future includes the following:

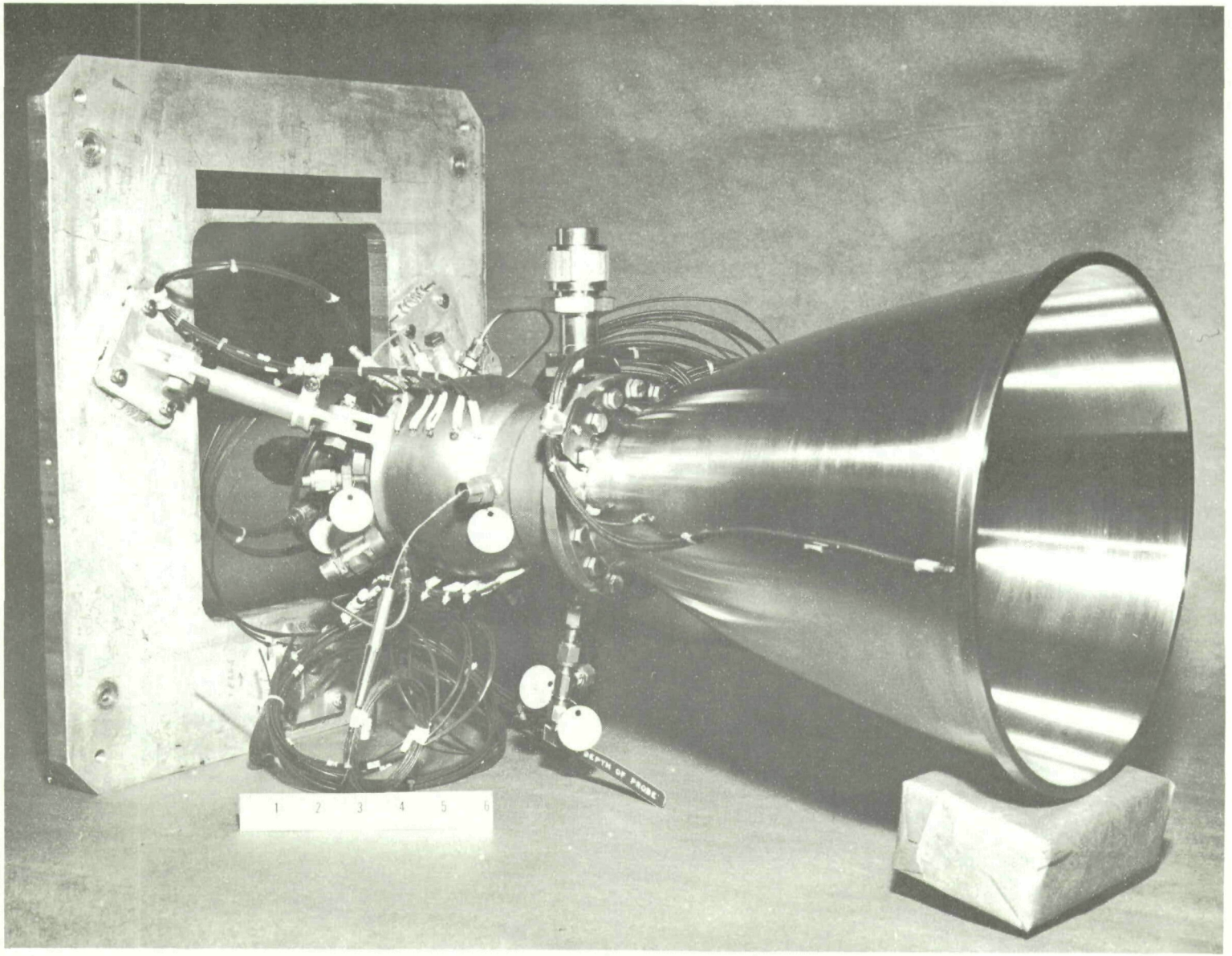
a. A 1000 lb thrust engine was assembled and a heat sink nozzle fabricated for altitude testing at the Marshall Space Flight Center (Figure 7).

b. Ignition testing with a 50 pound nominal thrust, Pc = 50 psia engine was in progress at BAC in a simulated altitude environment equivalent to 110,000 feet. The tests identified H_2 post run N_2 purge as a contributing factor in failure to achieve ignition. Elimination of the N_2 purge considerably improved the ignition reliability. Tests are planned to evaluate spark plug recess relative to the inner surface of the combustion chamber and to vary the geometry of the O_2 augmentation injection.

c. A regeneratively cooled nozzle section for sea level operation is being fabricated for tests at the 1500 lb nominal thrust level (Figure 8).

d. A 1500 lb thrust oxidizer injector providing annular O_2 injection with a pintle is being designed in an attempt to minimize ΔP and evaluate effect on performance.

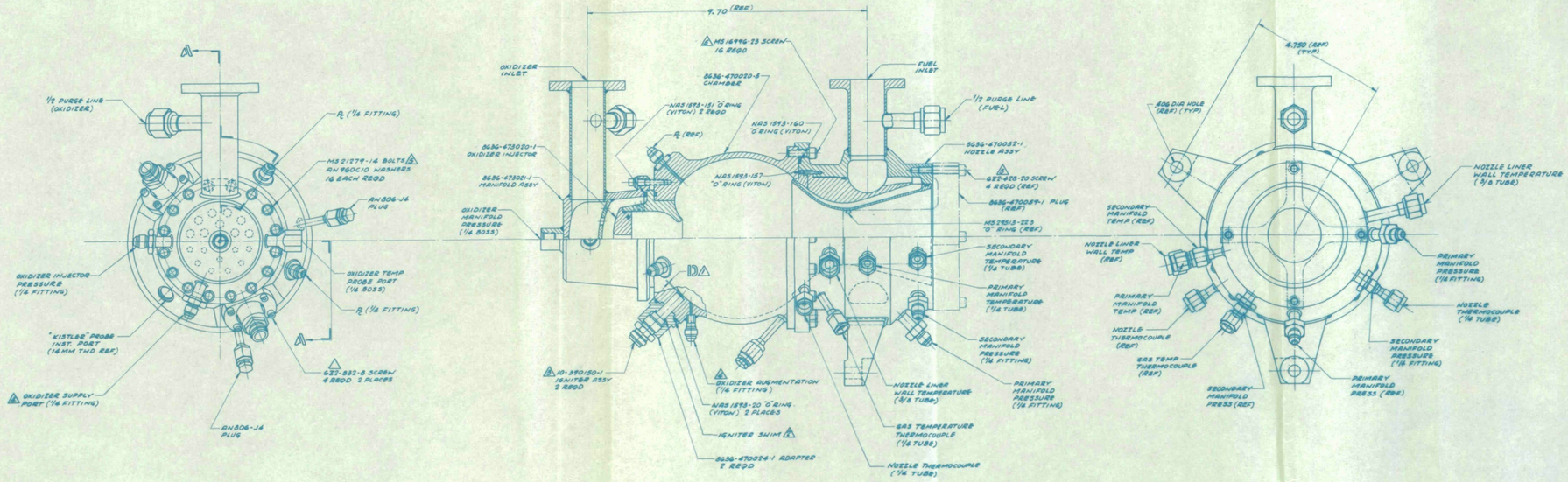
e. A 32L* combustion chamber is being fabricated for 1500 lb thrust testing. The fuel injector will be modified to move the point of fuel injection upstream closer to the plane of the O_2 entry to evaluate effect on performance and heat rejection.



1000 LB THRUST REVERSE FLOW ENGINE

FIGURE 7

8836-470080



QTY REQD	PART NO	CODE IDENT	PART OR IDENTIFYING NO	NOMENCLATURE OR DESCRIPTION	DRAWING OR SPECIFICATION	MATERIAL OR NOTE
2			10-370150-1	IGNITER ASSY		
8			GZZ-832-8	SCREW		
1			NAS 1593-160			
1			NAS 1593-157			
2			NAS 1593-131			
2			NAS 1593-20			
16			AN960C10	WASHER		
2			AN806-16	PLUG		
16			MS 21279-14	BOLT		
16			MS 16996-23	SCREW		
1			8836-473071-1	MANIFOLD ASSY		
1			8836-473020-1	OXIDIZER INJECTOR		
1			8836-470052-1	NOZZLE ASSY		
2			8836-470024-1	ADAPTER		
1			8836-470020-5	CHAMBER		
			8836-470080-1	THRUST CHAMBER ASSY		

PARTS LIST

△ TORQUE PER BPS 4018.

△ OXIDIZER SUPPLY PORT SHALL BE CONNECTED TO OXIDIZER AUGMENTATION PORTS WITH A 1/4" TUBE ASSY. LENGTH AND CONFIGURATION TO BE DETERMINED BY MOCK UP.

△ PURCHASE FROM BENDIX CORP. SCINTILLA DIV., SIDNEY N.Y.

△ THICKNESS OF IGNITER SHIM TO BE DETERMINED BY DIMENSION "D".

△ DIMENSION "D" TO BE DETERMINED BY ENGINEERING.

NOTES:

		THRUST CHAMBER ASSY, SEA LEVEL ENGINE	
		80070	8836-470080

Figure 8

f. Efforts will continue to relate the fuel and oxidizer injection design parameters to the combustion efficiency achieved.